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Aircraft Structures Technical Memorandum 527

AN OUTLINE OF A NUMERICAL SCHEME FOR CALCULATING TWO-DIMENSIONAL TIME LINEARISED TRANSONIC FLOW USING THE GREEN'S FUNCTION METHOD

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by

I. H. Grundy

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AN OUTLINE OF A NUMERICAL SCHEME FOR CALCULATING TWO-DIMENSIONAL TIME-LINEARISED TRANSONIC FLOW USING THE GREEN'S FUNCTION METHOD

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I. H. Grundy

SUMMARY

A numerical scheme is outlined for the calculation of two dimensional time-linearised transonic flow about an aerofoil or collection of components, using the Green's function method. Computed results are presented for various configurations for the simpler sub-problem of steady subsonic flow.

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CONTENTS

1. INTRODUCTION

This article presents an outline of a numerical scheme for calculating two-dimensional unsteady transonic flow about an aerofoil, or a collection of aerofoil components. The aerofoil components are assumed to be thin, and the effects of viscosity are neglected. Hence, the flow is assumed to be governed by the transonic small disturbance equation [1], [7].

The approach followed here is the Green's function (integral equation) method. This method, see [8-10] and also [2], was first introduced to compute unsteady subsonic aerodynamics in three dimensions. However, the method was extended in [14] to also handle transonic aerodynamics (see also [3]). The main advantage of the method is that it is easy to implement for complex geometries, since body-fitting grid systems are not required [5],[15].

In [14] and [3], the Green's function method involves "time-integration". That is, the flow about the aerofoil is calculated at each moment as the program steps forward in time. Alternatively, significant savings in computer time can be made by "time-linearising", i.e. by assuming that the aerofoil oscillates harmonically about some mean steady position ([5], [8], [9], and [11]).

The cost of "time-linearisation" is an inability to model highly non-linear phenomena such as shock appearance and disappearance, and large scale shock motion, such as that identified in [13]. However, the method is compatible with conventional subsonic (linear) flutter analysis, and significantly easier to program than "time-integration" methods.

As a first step towards a full "time-linearised" transonic capability, results are presented here for a simple sub-problem of two-dimensional unsteady transonic flow, namely the problem of two-dimensional steady subsonic flow. This preliminary problem, for which analytical solutions are available, provides a useful initial check on the method's accuracy and a pointer to possible problems in the full transonic formulation.

2. TRANSONIC PROBLEM

Under the assumptions that the fluid is isentropic and inviscid, the flow is irrotational and the aerofoil is thin and undergoes small unsteady motion, there exists a perturbation potential ϕ which must satisfy the unsteady small perturbation transonic equation [1], [7], namely

$$[(1 - M^2) - M^2(1 + \gamma)\phi_x]\phi_{xx} + \phi_{yy} - 2M^2\phi_{xt} - M^2\phi_{tt} = 0. \qquad (1)$$

where γ is the ratio of specific heats and the x-axis is in the streamwise direction. The Mach number M is assumed to be near, but less than unity, meaning that only "sub"-transonic flow will be considered here.

In order to reduce the computational time and complexity of the formulation, the governing equation is "time-linearised" as in [5] and [11]. That is, the potential is separated into a steady part and a small harmonically oscillating unsteady part. This leads to a non-linear boundary value problem for the

steady potential and a linear boundary value problem for unsteady potential (which is linked to the steady problem).

Equation (1) is "time-linearised" by setting $\phi = \phi_s + \phi_u$, where $\phi_u \ll \phi_s$, and keeping only the higher order terms. Accordingly, the steady potential ϕ_s must satisfy

$$(1 - M^2)\phi_{sxx} + \phi_{syy} = M^2(1 + \gamma)\phi_{sx}\phi_{sxx} \tag{2}$$

which can be rewritten, using the Prandtl-Glauert transformation $X = x$, $Y = \beta y$ (setting $\beta = \sqrt{1 - M^2}$), as

$$\phi_{sXX} + \phi_{sYY} = \sigma_s, \tag{3}$$

where the non-linear term σ_s is given by

$$\sigma_s = \frac{M^2(1 + \gamma)}{1 - M^2}\phi_{sX}\phi_{sXX}. \tag{4}$$

The unsteady potential ϕ_u must then satisfy the linear equation

$$(1 - M^2)\phi_{uxx} + \phi_{uyy} - 2M^2\phi_{uxt} - M^2\phi_{utt} = M^2(1 + \gamma)\frac{\partial}{\partial x}[\phi_{ux}\phi_{sx}]. \tag{5}$$

Application of the above Prandtl-Glauert transformation, plus the further substitution (assuming harmonic time dependence) of

$$\phi_u = \Phi_u e^{ik(t + M^2 X/\beta^2)} \tag{6}$$

into (5) leads to an equation for the new dependent variable Φ_u, namely

$$\Phi_{uXX} + \Phi_{uYY} + h^2\Phi_u = \sigma_u, \tag{7}$$

where $h = kM/\beta$ and the σ_u term on the right hand side (which is linear but dependent on the steady potential) is given by

$$\sigma_u = \frac{M^2(1 + \gamma)}{1 - M^2}\frac{\partial}{\partial X}[\phi_{sX}\frac{\partial}{\partial X}(\Phi_u e^{ikM^2 X/\beta^2})]. \tag{8}$$

It is not strictly necessary to use equations (3) and (4) to calculate the underlying steady flow field. An alternative, see [4] and [5] and elsewhere, is to use the steady solution from a much simpler formulation than the transonic small perturbation equation, thus making computer time savings.

Conversely, the steady flow field could be generated by a more complicated equation such as the full potential equation. The advantage of this approach over the use of the small disturbance equation is that shock strength and position are modelled more realistically.

An attractive feature of using the full-potential equation for the steady flow field is that this equation can be solved with relatively little extra cost over the small disturbance equation. Solving the steady full potential equation simply means using a different non-linear term σ_s, as in [15].

The method of solution of both the steady and unsteady problems is based on the Morino integral equation method [8], in which Green's Formula is used to construct an integral equation for the velocity potential ϕ, where ϕ represents ϕ_s in the steady case, and Φ_u in the unsteady case.

Green's formula for both the steady and unsteady equations relates the potential ϕ_P at a point P in the flow domain to an integral of the potential over the aerofoil boundary C_S, the wake contour C_W, any shock discontinuities C_D, and the contour at infinity C_∞. It has the form

$$E\phi_P = \int_C (G\frac{\partial\phi}{\partial N} - \phi\frac{\partial G}{\partial N})\, d\ell + \int_F G\sigma\, dS \tag{9}$$

where $C = C_S \cup C_W \cup C_D \cup C_\infty$ (as shown in Figure 1), $\mathbf{N}$ is the inward normal to the fluid in the Prandtl-Glauert space, and σ represents either σ_s or σ_u.

The domain function E is defined by

$$E = \begin{cases} 1, & \phi_P \text{ within flow field} \\ \frac{1}{2}, & \phi_P \text{ on the boundary} \\ 0, & \phi_P \text{ outside flow field.} \end{cases} \tag{10}$$

For the steady problem

$$G = \frac{1}{2\pi}\log R \tag{11}$$

(where R is the distance to the singularity) is an appropriate Green's function, while for the unsteady problem

$$G = \frac{1}{4} i H_0^{(2)}(hR) \tag{12}$$

(where $H_0^{(2)}$ is the Hankel function of the second kind of order zero) is an appropriate Green's function.

Using the far-field expansions in [1] and [6], and the Green's functions above, it can be shown that at infinity the contribution to the righthand side of (9) is at most a constant. Hence, the boundary integral over C_∞ may be omitted. Also, the integral over the shock discontinuity C_D can be combined with the volume integral in such a way that there is no explicit contribution to ϕ_P (as a result of the shock jump conditions) [6], [14]. Since the new integral equation generated this way can be shown to differ very little from the original integral equation "at the numerical algorithm level" [14] when discretised, the original equation can be used ignoring altogether the presence of shocks, which are said to be "captured". As in [12], artificial viscosity can be used when evaluating the field integrals to guarantee the absence of (i.e. smooth out) shock waves.

A primary advantage of the integral equation approach is that no complex, body-fitting grid generation scheme is needed within the flow field (such as would be required in a finite difference method). Most of the effort in grid generation is needed at the aerofoil boundary, which is approximated by small elements (line segments in 2D).

Although the flow field F must still be discretised in order to evaluate the righthand side of equation (9), a cartesian grid (consisting of rectangular panels) is sufficient for this purpose, and this grid need not be fitted to the shape of the aerofoil contours. In fact, this grid need only be generated where the transonic terms σ_s and σ_u are significant, i.e. near the aerofoil (see [5],[15]). Also, both the steady and unsteady calculations can take place on the same grid.

3. PROPOSED NUMERICAL SCHEME – AN OUTLINE

The numerical scheme proposed here involves the two stages described above, namely, initial calculation of the steady flow field, followed by calculation of the unsteady flow field. Each stage uses the same grid, but with differing Green's functions and σ terms. The calculation scheme for each stage is the same and proceeds as follows:

Step 0 Evaluate $\phi = \phi_C$ on the aerofoil contour using a normal (subsonic) panel method, that is, setting the field source strength σ to zero. The aerofoil surface is divided into panels and the unknown ϕ_C is approximated in some fashion, e.g. as being constant on each panel. The known quantity $\frac{\partial\phi}{\partial N}$ is also approximated in some consistent fashion. The equation for ϕ_C,

$$E\phi_C = \int_C (G\frac{\partial\phi}{\partial N} - \phi_C\frac{\partial G}{\partial N})\, d\ell \tag{13}$$

where $C = C_S \cup C_W$ becomes a matrix equation for the discrete ϕ_C values.

Step 1 Evaluate $\phi = \phi_F$ in the flow field F using the values of ϕ_C calculated at the previous step, and the current value of σ. The flow field is discretised into panels on which, for example, ϕ_F is constant. The discrete version of

$$E\phi_F = \int_C (G\frac{\partial\phi}{\partial N} - \phi_C\frac{\partial G}{\partial N})\, d\ell + \int_F G\sigma\, dS \tag{14}$$

yields ϕ_F at each field panel.

Step 2 Evaluate the field source strength σ by applying finite differences to formulas (4) or (8). The expression for σ is non-linear for the steady problem, while for the unsteady problem it is linear but depends on the calculated steady solution.

Exit iteration loop if field source strengths have converged.

Step 3 Evaluate $\phi = \phi_C$ on the aerofoil boundary using the transonic panel method (with field source term). This step echoes Step 0 and involves solution of the linear matrix equation arising from discretisation of

$$E\phi_C = \int_C (G\frac{\partial\phi}{\partial N} - \phi_C\frac{\partial G}{\partial N})\, d\ell + \int_F G\sigma\, dS \tag{15}$$

Go to Step 1

Stability and convergence have not been discussed here and are not guaranteed. However, in [15] encouraging results are given for the convergence of a similar natural iteration scheme. For subcritical (shock-free) flow, convergence to acceptable accuracy occurs within 5 iterations, while for supercritical flows (with shocks) convergence occurs within 30 iterations.

4. SUBSONIC PROBLEM

The subsonic problem is a subset of the transonic problem, corresponding to Step 0 above, in which the field source strength is zero and the steady Green's function (11) is used. This problem is considered first in order to identify areas of possible difficulty in the panelling and/or evaluation of the influence coefficients, and to provide the programming framework for later extensions to steady transonic flow (involving non-linear terms and iteration) and unsteady flow (more complicated Green's function).

The appropriate boundary value problem for the steady potential ϕ is to satisfy equation (2) with associated boundary conditions

$$\frac{\partial \phi}{\partial n} = -\mathbf{i} \cdot \mathbf{n}. \tag{16}$$

on the aerofoil boundary. As seen already, the Prandtl-Glauert transformation puts the transonic small perturbation equation into the form given in equations (3) and (4), while the boundary conditions (16) become

$$\frac{\partial \phi}{\partial N} = -\frac{\mathbf{i} \cdot \mathbf{N}}{\beta^2}. \tag{17}$$

Note that $\mathbf{N}$ is the normal to the aerofoil in the transformed flow domain, and that higher order terms have been neglected here, consistent with the linearisation of the governing equation.

The integral equation (13) for ϕ, obtained by application of Green's formula, is solved by the method of collocation. The aerofoil contour C_S is assumed to consist of N_t discrete panels P_j whose end points are (x_{j-1}, y_{j-1}) and (x_j, y_j), with equation

$$x = a_j y + b_j. \tag{18}$$

(In [2] the wing surface is fitted exactly and the boundary integrals are carried out numerically, whereas in the Morino method an approximate geometry is used and the integrals are usually calculated analytically, especially near singularities.) The collocation points (ξ_i, η_i) are taken to be the N_t panel midpoints, with associated values of the potential $\phi_i = \phi(\xi_i, \eta_i)$.

At the ith collocation point we have

$$\pi \phi_i = \int_{C_S} \log R_i \frac{\partial \phi}{\partial N} \, d\ell - \int_{C_S} \phi \frac{\partial}{\partial N}(\log R_i) \, d\ell - \int_{C_W} \phi \frac{\partial}{\partial N}(\log R_i) \, d\ell \tag{19}$$

where $R_i = \sqrt{(x - \xi_i)^2 + (y - \eta_i)^2}$.

Assuming ϕ is constant on each panel, and treating the wake boundary as a straight line emanating from the trailing edge across which there is no pressure jump, we have

$$\begin{aligned}\pi\phi_i &= \sum_j \int_{P_j} \log R_i \left(-\frac{\mathbf{i}\cdot\mathbf{N}}{\beta^2}\right) d\ell \\ &- \sum_j \phi_j \int_{P_j} \frac{\partial}{\partial N}(\log R_i)\, d\ell \\ &- \Delta\phi_{te} \int_{C_W} \frac{\partial}{\partial N}(\log R_i)\, d\ell\end{aligned} \tag{20}$$

for $i = 1, 2, \ldots, N_t$, where ϕ_{te} is the value of ϕ at the trailing edge. This leads to

$$\begin{aligned}\pi\phi_i &= \sum_j \int_{P_j} \frac{1}{\beta^2} \log R_i\, dy \\ &- \sum_j \phi_j [-\int_{P_j} \frac{x-\xi_i}{R_i^2}\, dy + \int_{P_j} \frac{y-\eta_i}{R_i^2}\, dx] \\ &- \Delta\phi_{te} \int_{x_{te}}^{\infty} \frac{y_{te}-\eta_i}{R_i^2}\, dx.\end{aligned} \tag{21}$$

for $i = 1, 2, \ldots, N_t$, where (x_{te}, y_{te}) are the coordinates of the trailing edge, and N_{te} is the panel number of the (upper) trailing edge panel. Evaluating the integrals we have, finally

$$\pi\phi_i = \sum_{j=1}^{N_t} B_{ij} - \sum_{j=1}^{N_t} A_{ij}\phi_j - \sum_{j=N_{te}}^{N_{te}+1} W_{ij}\phi_j \tag{22}$$

for $i = 1, 2, \ldots, N_t$, where

$$\begin{aligned}B_{ij} = \frac{1}{(a_j^2+1)\beta^2} \Big[&(y-\eta_i + a_j(x-\xi_i))(\log R_i - 1) \\ &+ (a_j\eta_i + b_j - \xi_i)\arctan(\frac{y-\eta_i+a_j(x-\xi_i)}{a_j\eta_i+b_j-\xi_i})\Big]_{y_{j-1}}^{y_j},\end{aligned} \tag{23}$$

$$A_{ij} = \begin{cases} \Big[-\arctan(\dfrac{y-\eta_i+a_j(x-\xi_i)}{a_j\eta_i+b_j-\xi_i})\Big]_{y_{j-1}}^{y_j} & \text{for } a_j \neq \infty \\ \Big[\arctan(\dfrac{x-\xi_i}{y_j-\eta_i})\Big]_{x_{j-1}}^{x_j} & \text{for } a_j = \infty, \end{cases} \tag{24}$$

and

$$W_{ij} = \pm \lim_{X\to\infty} \Big[\arctan(\frac{X-\xi_i}{y_{te}-\eta_i}) - \arctan(\frac{x_{te}-\xi_i}{y_{te}-\eta_i})\Big], \quad j = \{ {N_{te} \atop N_{te}+1} \tag{25}$$

The above set of simultaneous equations is now solved for the unknowns ϕ_j.

In the unsteady case, the Green's function is so complicated that the influence coefficients can only be evaluated numerically. Numerical experiments have

determined that for the steady problem a four point Gaussian rule is sufficiently accurate for use in evaluation of the influence coefficients, except near the collocation point where it becomes necessary to extract the singularity and evaluate that part separately (analytically).

For interfering aerofoils/components the wake terms W_{ij} need very careful treatment, especially when the idealised (i.e. straight) wake of one aerofoil component intersects with another component downstream. An interpretation used successfully here is that the arctan terms in W_{ij} represent the angles β and α respectively, shown in Figure 2. These angles change continuously as the point (ξ_i, η_i) crosses the wake (even though the arctan function jumps). As a result, $\pm 2\pi$ must be added to W_{ij} where necessary to retain continuity.

Once the potential has been obtained, an obvious method for calculating the pressure coefficient would be to differentiate equation (22) with respect to ξ_i and η_i and use these derivatives in all calculations. This works quite well within the flow field (provided the factor of π is replaced by 2π in (22)). However, it also leads to large errors near the aerofoil because of the assumption that the potential varies as a step function. The function given by the right hand side of equation (22) becomes almost step-like between collocation points, causing the calculated derivatives to be useless near these points, i.e on the aerofoil boundary.

Since direct calculation is impossible, an alternative is to calculate derivatives by finite differences. This is done using the collocation points, at which ϕ is known best.

Returning to the original unscaled co-ordinates, the pressure coefficient C_p is calculated from the following formula, applicable to thin geometries:

$$C_p = -2[\mathbf{i} \cdot \mathbf{q} + \frac{1}{2}\mathbf{q} \cdot \mathbf{q}] + M^2(\mathbf{i} \cdot \mathbf{q})^2 \tag{26}$$

where $\mathbf{q} = \nabla \phi$ is the perturbation velocity. On the aerofoil boundary we write

$$\mathbf{q} = q_s \mathbf{s} + q_n \mathbf{n} \tag{27}$$

where $\mathbf{s}$ and $\mathbf{n}$ are the tangential and normal vectors respectively. The pressure coefficient can thus be rewritten as

$$\begin{aligned} C_p = & -2[q_s(\mathbf{i} \cdot \mathbf{s}) + \frac{1}{2}(q_s^2 - q_n^2) \\ & -\frac{1}{2}M^2(q_s^2(\mathbf{i} \cdot \mathbf{s})^2 - 2q_s q_n^2(\mathbf{i} \cdot \mathbf{s}) + q_n^4)], \end{aligned} \tag{28}$$

which has the advantage that the magnitude of the normal velocity component $q_n = -\mathbf{i} \cdot \mathbf{n}$ is known exactly from the boundary conditions, while the magnitude of the tangential velocity q_s is easily approximated by finite differences at the boundary. Thus finally, the following formula applies for the pressure coefficient at the ith collocation point:

$$\begin{aligned} C_{pi} = & -2\Big[\frac{\Delta\phi_i \Delta\xi_i + \frac{1}{2}(\Delta\phi_i)^2 - \frac{1}{2}(\Delta\eta_i)^2}{(\Delta\xi_i)^2 + (\Delta\eta_i)^2}\Big] \\ & + M^2\Big[\frac{(\Delta\phi_i)^2(\Delta\xi_i)^2 - 2\Delta\phi_i\Delta\xi_i(\Delta\eta_i)^2 + (\Delta\eta_i)^4}{[(\Delta\xi_i)^2 + (\Delta\eta_i)^2]^2}\Big] \end{aligned} \tag{29}$$

where

$$\Delta\varphi_i = \varphi_{i+1} - \varphi_{i-1}, \tag{30}$$

$$\Delta\eta_i = \eta_{i+1} - \eta_{i-1} \tag{31}$$

and

$$\Delta\xi_i = \xi_{i+1} - \xi_{i-1}. \tag{32}$$

5. RESULTS AND DISCUSSION

Results have been computed for the case $M = 0$ for three test configurations, namely

1. A circle with unit radius centered at the origin.
2. A NACA0012 aerofoil at zero angle of attack.
3. Williams configuration B [16] (aerofoil plus flap at 10 deg).

Figures 3 to 5 are plots of $-C_p$ vs chordwise position for these three configurations. The exact incompressible solutions are denoted by lines and numerical solutions are denoted by asterisks.

The agreement in all cases is excellent. The agreement at the trailing edges in configuration 3 (Figure 5) is surprisingly good in view of the assumption that each wake is a straight line emanating from the trailing edge.

6. CONCLUSION

In this article the outline of a numerical scheme for the calculation of two-dimensional unsteady transonic aerodynamics has been presented. Excellent results have been obtained for the first step towards implementing this scheme, that is, for the special case of steady subsonic aerodynamics.

Despite the apparent simplicity of the outlined scheme, implementation for unsteady and transonic aerodynamics will still involve significant leaps in difficulty. For unsteady flow, a more complicated Green's function will mean that the integrals involved can no longer be evaluated in closed form. Difficulties are also anticipated in the application of the Kutta condition at the trailing edge. For transonic flow the addition of field source terms, and the consequent need for an iterative numerical scheme, will lead to demands on computer time and storage. Problems with stability and convergence of the numerical scheme are also a possibility.

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FIGURES

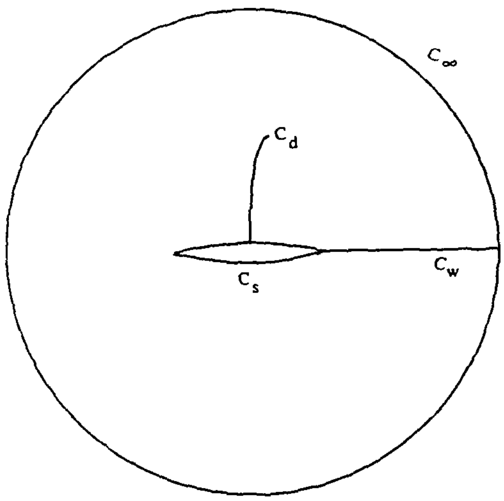

Figure 1. *Flow domain showing important contours.*

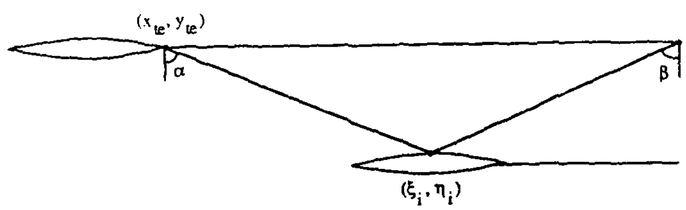

Figure 2. *Diagram showing angles between the trailing edge, the wake and collocation point.*

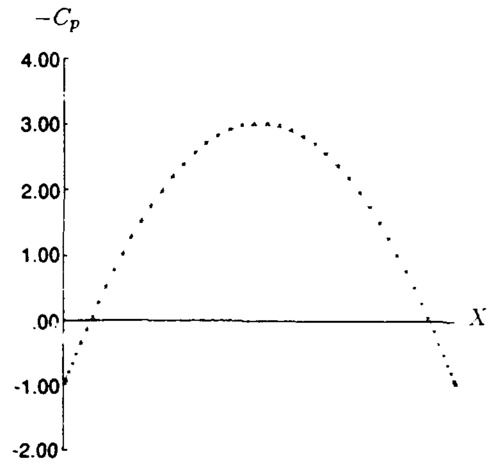

Figure 3. *$-C_p$ vs X for flow around a circular cylinder – Comparison of numerical results $*$ with the exact solution — .*

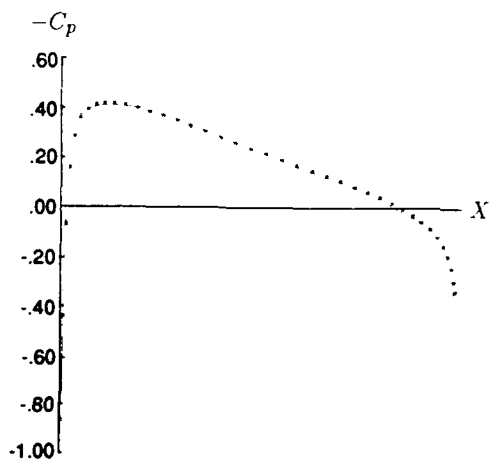

Figure 4. *$-C_p$ vs X for flow around a NACA0012 aerofoil at zero incidence – Comparison of numerical results $*$ with the exact solution — .*

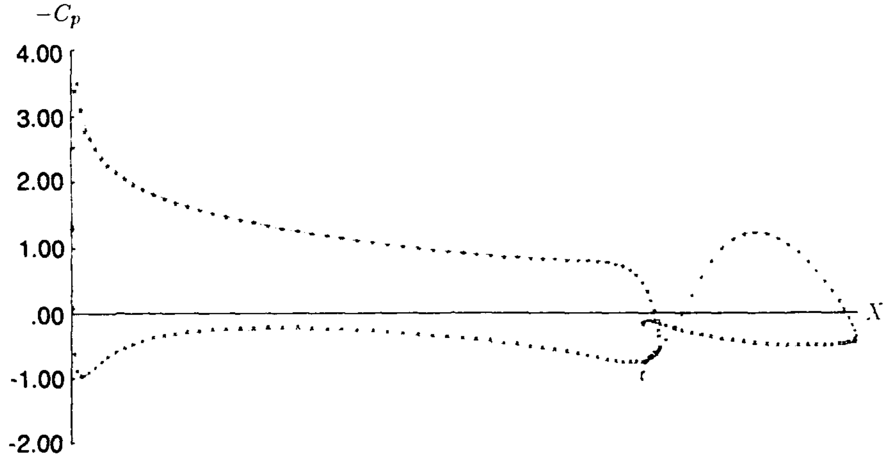

Figure 5. $-C_p$ *vs* X *for flow around Williams configuration B – Comparison of numerical results* $*$ *with the exact solution* — .

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A numerical scheme is outlined for the calculation of two dimensional time-linearised transonic flow about an aerofoil or collection of components, using the Green's function method. Computed results are presented for various configurations for the simpler sub-problem of steady subsonic flow.

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